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Preliminary Structural Sizing of a Modular Microsatellite Based on System Engineering Considerations

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Abstract
In this paper a system engineering approach toward the preliminary design of the structure subsystem of a microsatellite has been introduced. This work is on the context of the UPMSat-2 UNION, a university class small satellite under design at Universidad Politécnica de Madrid (Spain), whose aim is to introduce a reliable platform for different types of missions and payloads. Launch loads are pre assumed based on the data provided by Ariane Structure for Auxiliary Payload 5, ASAP5 launcher with the condition of launching the satellite as a secondary payload. The satellite main configuration includes four trays. One tray is assumed for mission payload and the three others for accommodating the instruments related to the other subsystems. The structure subsystem mass is calculated from the analytical formulas in order to fulfil the static constraints and frequency requirement. The main purpose of the work is finding out in a simple way the primary structure behaviour before entering the FEM and more detailed analysis. From system engineering approach it is desirable if the satellite platform can offer as much as volume and mass for the payload, because it results to providing more diversity for payload types. Based on this, different satellite configurations adjustable to the envelope of piggyback launch are considered and their provided payload volume in comparison with the structural mass required to stand the static loads and fulfil frequency requirements is investigated.

1 Introduction

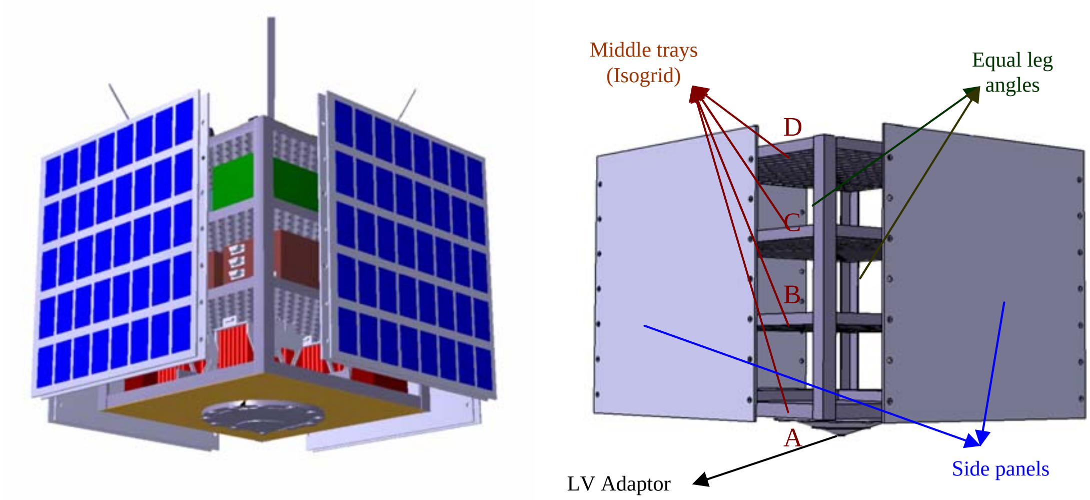
For more than 25 years, small satellites have been the best choice for universities and research institutes toward starting space technology development. These projects are attractive for educational purposes because of important reasons such as their small mass and dimensions which enable them be launched as an auxiliary payload that result to less cost and test requirements. Furthermore, with Commercial Off The Shelf (COTS) components it is possible to design them with different types of missions like store and forward communication or rough remote sensing in a short time schedule and less complexity than bigger space projects. On the other hand, these kinds of projects create a framework in which professors and students can learn and improve their existing knowledge of the ins and outs of aerospace engineering with a relatively low cost programs. Moreover, the global economic recession has created an ‘opportunity space’ for small satellite systems to capture a broader acceptance also in industry which means there will be potential demands in outer market for the efforts taken in universities on small satellite projects. (Alale et al. [1], Swartwout [10])

In this context, the Instituto Universitario de Microgravedad “Ignacio Da Riva” (IDR/UPM) from the Universidad Politécnica de Madrid (Spain) is developing the UPMSat-2 UNION a microsatellite being designed and built by professors, graduate and undergraduate students. This project is a part of the IDR/UPM space activities and it is based on the previous experience of the UPMSat-1 launched on July 7, 1995 by Ariane IV (Sanz-Andres et al. [7]). In addition to the educational purposes, the satellite is foreseen to be a versatile and low cost satellite platform, which is compatible for the development of scientific and technological missions within the university environment. UPMSat-2 UNION will be launched to a polar orbit at near 600 km of altitude as a secondary payload and its launching date is scheduled in 2014.

This work focus is to introduce a rapid and simple sizing tool for the preliminary structural calculations of a modular microsatellite which also cover the UPMSat-2 UNION structural characteristics. An analytical simple and parametric mathematical model of the satellite primary structure has been developed. The loads imposed on satellite during the launch are extracted based on the mechanical requirements of the launcher and then, the effect of this load condition is analysed on primary structure elements. This effect can be investigated through maximum stress, maximum deflection and buckling strength of the primary structures. The dynamic behavior of the primary structure is also described with a simple model of concentrated mass and equivalent spring, which is a function of the geometrical parameters, mass properties and materials to build the satellite. The purpose of this model is to check the frequency requirements of the launcher. Both models have been implemented in software in order to apply for different cases. Five different cases for satellite mass distribution and dimension are considered based on the previous projects. Regarding system engineering aspect, for each case the volume which is provided for the payload and its structural mass to withstand the loads and fulfill frequency requirements is investigated.

2 Satellite configuration

The satellite is composed of four middle trays, A, B, C, and D positioned from the attachment point to the launch vehicle up to the top of the satellite respectively, as shown in the Figure 1. The bottom tray (Tray A) serves as a link with the separation system and contains the batteries and energy management electronics. Tray B houses most of the electronic components, as the command and data handling and communication subsystems. Tray C gives support to the payload and Tray D will accommodate the necessary external sensors as well as communication antennas. Aluminum alloy 7075 is selected for the material. The primary structural elements consist of flat plates for side panels and equal leg angles as main frame and isogrid plates for middle trays. Equal leg angles are selected to have 3 cm leg length and leg thickness vary between 0.5 and 2.75 mm. Also, the side panel thickness is selected in range of 0.75 and 3 mm.



(a) (b)

Figure 1: a) Satellite configuration, b) Satellite primary structures: 4 side panels, 4 middle trays and 4 equal leg angles

Based on the previous experience on UPMSat-1 (Sanz-Andres et al. [8]), and general design estimation relationships (Ravanbakhsh et al. [6]) a preliminary estimation of different subsystems mass are indicated in Table 1.

Table 1: Subsystems mass percentage of total mass and their position in satellite

Subsystems	Percentage of satellite total mass [%]	Position
Electrical Power (EPS)	20	Tray A
Attitude Determination and Control (ADCS)	10	Tray A, B, C, D
Communications (TT&C)	5	Tray B (boards), Tray D (antennas)
On board Data Handleing(OBDH)	4	Tray B
Thermal Control (TC)	1	Distributed
Payload (PL)	40 (foreseen)	Tray C
Structure (STR)	20 (foreseen)	Distributed

According to generic approach to the COTS components a conservative distance of 12.5 cm between tray A, tray B as well as tray B and tray C is assumed in order to provide proper volume to accommodate the different subsystems.
The mass distribution between different trays for five different satellite sizes is obtained in Table 2. The satellite total mass and overall dimensions should satisfy the piggy back launch envelope. In this regard a density of 400 kg/m^3 is considered for the approach based on some educational university class satellites data. (Thyagarajan et al. [12], Swartwout [10]).

Table 2: Mass distribution in each tray, A, B, C, D (expressed in kg) for different satellite dimensions

Satellite dimension	40×40×40 cm	45×45×45 cm	50×50×50 cm	55×55×55 cm	60×60×60 cm
Satellite mass [kg]	25.50	36.50	50.00	66.50	86.50
Tray A	6.60	9.60	13.35	18.00	23.50
Tray B	3.25	5.20	7.35	10.15	13.50
Tray C	11.70	16.90	23.40	31.20	40.75
Tray D	1.95	2.80	3.90	5.15	6.75

2.1 Satellite structure mass

The total mass of primary structures is sum of the mass of four elements; 4 equal leg angles acting as the main frame, 4 plates as side panels, 4 isogrid plates as the middle trays, A, B, C and D and also launch vehicle adaptor which will remain to the satellite after injection in to the orbit. The Launch vehicle adaptor mass in estimated to be 2 kg (Sanz-Andres et al. [8]). The mass of other elements is calculated based on the static analysis as well as frequency requirement of the satellite.
The frequency requirement for middle trays is determined by the rectangular plate vibration analysis. The first five non dimensional natural frequencies using Bessel functions for fully simply supported square plate and for fully clamped square plate is considered from (Liu et al. [5]). The non dimensional natural frequency, ka of the rectangular plate is used in which a is the side length of the plate and $k^4 = \omega^2 \rho t / D$. In this relation, ω is the natural frequency, ρ is the mass density of the plate, t is the thickness of the plate, and D is the plate bending rigidity $D = Et^3/12(1-\nu^2)$. Based on these relations the minimum natural frequency of the middle trays is calculated and the mass of acceptable ones according to the frequency requirement are obtained. Also because there is no certainty of boundary conditions of middle trays inside the satellite the average of frequencies in simply supported and clamped supported responses are assumed at this stage of the design.
In middle trays structural design, it is planned to use isogrid structure plates instead of honeycombs or monocoque plates. This is because of their less weight in comparison to monocoque plates and less cost and complexity in construction compared to honeycombs. To fulfill this objective by using (Isogrid Design Handbook [4]), the equivalent isogrid plate mass is determined for all five different satellite size cases, as shown in Figure 2.

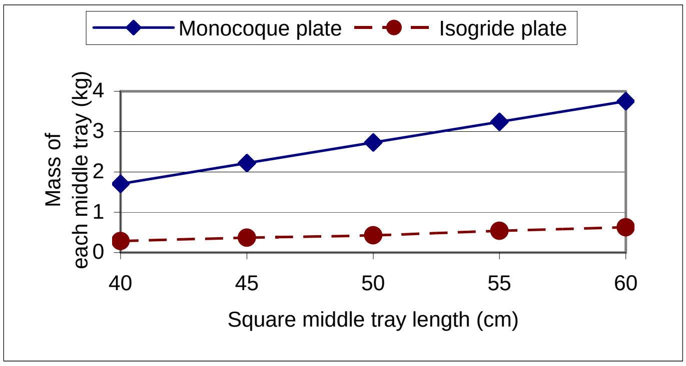


Figure 2: Mass of acceptable middle trays (monocoque and isogrid) regarding frequency requirement

3 Static analysis

Physically the bottom of the satellite where attached to the launcher can be considered as clamped support. So, for static analysis, the satellite has been assumed as a cantilever beam with the uniform mass distribution. Also, the lateral load is assumed to act on the satellite as uniform load along the longitudinal axis. The design loads has been determined by considering safety factor of 1.5 and uncertainty factor of 1.25, (Sarafin [9]).

In the static analysis after reviewing the results it seems that the maximum stress and deflection are not so critical and buckling should be considered as design criteria in the static analysis.

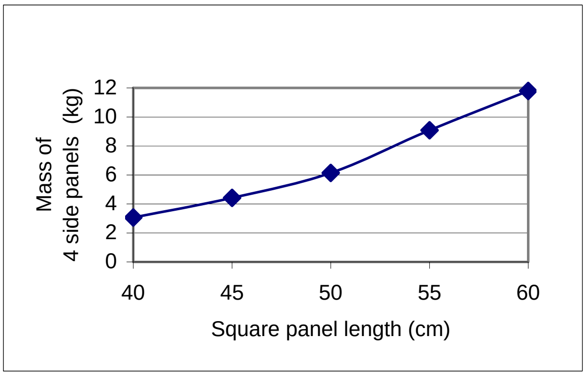
The loads are extracted based on the information of ASAP5 launcher which the launcher of UPMSat-2 has not been confirmed at this stage. According to (ASAP5 User's Manual [2]), the quasi static acceleration in longitudinal direction is (- 7.5 g / + 5.5g) and in lateral direction is (± 6 g).

3.1 Buckling

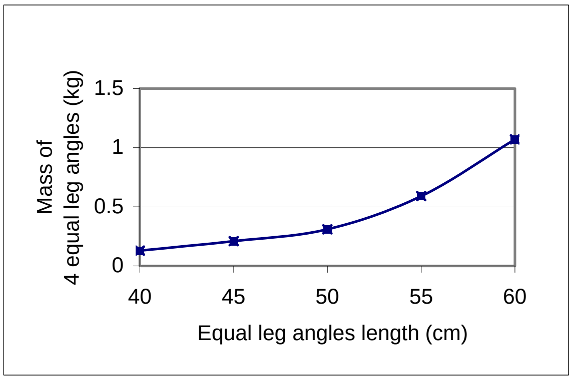
The buckling analysis has been done separately for equal leg angles profiles and plates which act as side panels. In the case of equal leg angles the normal loads which are exerting on each of equal leg angles is from two sources, direct longitudinal force and the normal force comes from the bending moment produced by the lateral loads. In case of side panels, for each of two in front side panels, two case of buckling can be assumed; buckling from compression stress and buckling from combined shear and longitudinal stress. In each case the critical buckling stress and its related design M.S. (Margin of Safety) can be determined from (Bruhn [3]).

3.2 Static analysis results

For all of the elements combinations the above static calculations have been done and the results are shown in the following figure for the cases with Buckling M.S. of 50%.



(a)



(b)

Figure 3: a) Mass of 4 side panels, b) Mass of 4 equal leg angles

4 Natural frequency determination

In order to determine the minimum natural frequency of the satellite simple mass-spring models has been assumed in longitudinal and lateral directions. The equivalent stiffness for each part has been considered properly and the mass on each tray is assumed based on different subsystems mass distribution. According to (ASAP5 User's Manual [2]), the fundamental frequency in the longitudinal axis should be grater than 90 Hz and the fundamental frequency in the lateral axis should be greater than 45 Hz.
For longitudinal direction, in order to determine the minimum natural frequency, a 4 DOF (Degree Of Freedom) mass-spring model is assumed as the representative of the satellite, Figure 4. Each tray is considered as a concentrated mass and the stiffness between trays are estimated as $k_{eq} = EA/l$ which is calculated for both side panels and equal leg angles. In this relation, E is Young's modulus, A is the cross section area of elements, and l is the length of elements. Also, the separation system longitudinal stiffness is calculated based on its bolt longitudinal stiffness, (Thomson [11]).

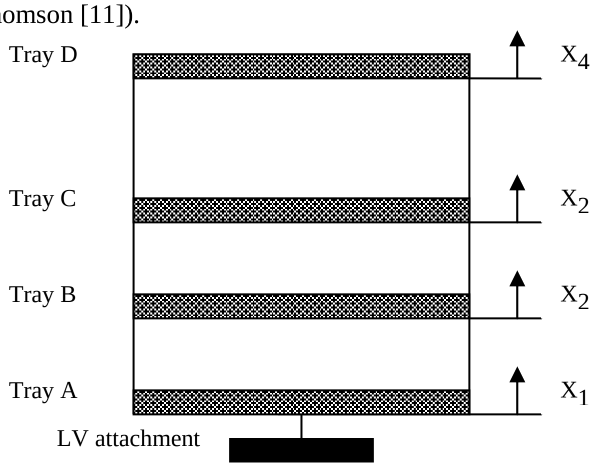


Figure 4: 4 DOF model assumed for satellite in longitudinal direction

In lateral direction, a 3DOF mass-spring is assumed, Figure 5. In this model, tray A and LV adaptor are considered to be completely clamped together and to the launcher and just tray B, C, and D are considered as separate concentrated mass. The equivalent lateral stiffness for equal leg angles is estimated by $k_{eq} = 12EI/l^3$, in which, E is Young's modulus, I is the second moment of area of elements, and l is the length of elements.
And for the two shear tolerating side panels as $k_{eq} = F_r Gdt/l$ where F_r is reduction factor because of not complete rigid boundary conditions of the side panels, G is Shear modulus, d is the lateral length of the plates, t is plate thickness, and l is the longitudinal length of the plates, (Thomson [11]).

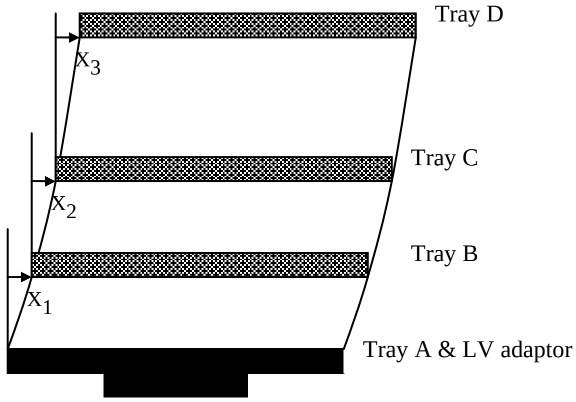


Figure 5: 3 DOF model assumed for satellite in lateral direction

5 Sizing tool validation

The sizing tool which has been used in the analysis results to estimation of satellite structure mass according to pre assumed structural mass allocation equal to 20% of the satellite total mass. In order to check the validity of the sizing tool, the obtained results for the mass of satellite primary structures, which satisfy the static and frequency requirements, are compared to this allocation as seen in the following Table 3.

Table 3: Satellite primary structures mass obtained from sizing tool

Satellite dimension	40×40×40 cm	45×45×45 cm	50×50×50 cm	55×55×55 cm	60×60×60 cm
Satellite mass [kg]	25.50	36.50	50.00	66.50	86.50
Mass of 4 equal leg angles [kg]	0.13	0.21	0.46	0.83	1.49
Mass of 4 side panels [kg]	3.06	4.42	6.14	9.08	11.79
Mass of 4 isogrid middle trays [kg]	1.14	1.48	1.71	2.16	2.50
Total structural mass [kg]	4.33	6.11	8.31	12.07	15.78
Percentage of total mass [%]	16.91	16.76	16.32	17.78	17.78
Percentage of difference with foreseen value [%]	3.09	3.24	3.68	2.22	2.22

As seen for all cases the difference between the obtained and forssen value of structure mass is less than 4%. Also by considering the mass of secondary structural components the pre assumed structural mass and the obtained results seem to be more in accordance with each other. All over, it can be concluded that at this stage of the design the results of the sizing tool seems satisfactory.

6 Results

The static and frequency analysis is conducted for all five satellites with different mass and dimensions and the results are indicated in the Table 4.

Table 4: Static and frequency analysis results for different satellite dimensions

Satellite dimension	40×40×40 cm	45×45×45 cm	50×50×50 cm	55×55×55cm	60×60×60 cm
Satellite mass [kg]	25.50	36.50	50.00	66.50	86.50
Longitudinal frequency [Hz]	169.50	140.50	119.10	102.8	90.00
Lateral frequency [Hz]	88.90	88.10	84.80	93.50	80.10
Available mass for the payload [kg]	10.24	14.58	20.00	26.62	34.56
Available volume for payload [cm^3]	16000	30375	50000	75625	108000

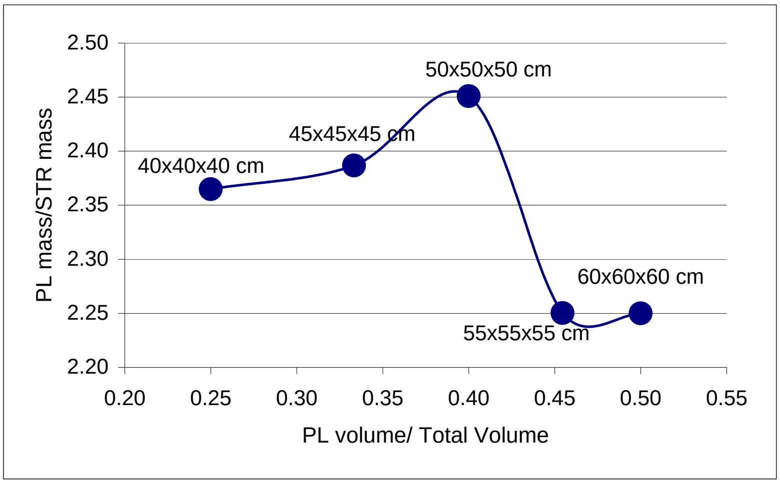


Figure 6: PL mass and volume ratios for five satellite dimensions

Regarding system engineering aspect it is desirable if the satellite can provide as much as possible volume to the payload. It is apparent that by increasing the satellite dimension the provided volume for the payload also will increase. But this increase in payload volume how is related to the payload mass and structural mass? In Figure 6, the ratio between PL mass and STR mass has been plotted versus the ratio between PL volume and satellite total volume. Based on the mission and the importance of PL volume or PL mass the whole size of the satellite can be chosen. According to the obtained results the satellite with 50×50×50 cm dimensions is optimum regarding to allocate volume to the payload with smaller payload to structure mass ratio.

7 Conclusion

Satellite design at any class is a complex process. System engineering approach toward the whole design in different aspects is important at early design stages in order to establish the optimum decisions based on the whole objective of the satellite mission. The present work is done in the context of UPMSat-2 UNION, a microsatellite under design by professors, graduate and undergraduate students at the Instituto Universitario de Microgravedad “Ignacio Da Riva” (IDR/UPM) from the Universidad Politécnica de Madrid (Spain). A basic system engineering approach toward the preliminary structural sizing of a modular microsatellite is conducted. The main aim is to evaluate different possible dimensions and mass distribution according to the compatibility to piggy back launch. Launch loads and frequency requirements are pre assumed based on the data provided by Ariane Structure for Auxiliary Payload 5, ASAP5 launcher. The satellite main configuration includes four trays. One tray is assumed for mission payload and the three others for accommodating the instruments related to the other subsystems. A sizing tool based on analytical simple and parametric mathematical model of the satellite primary structure has been developed. Five satellites with different size and mass distribution have been investigated. The evaluation of the structural mass toward provided mass and volume for the payload is done. An acceptable estimation has been reached and the initial step toward sizing the UPMSat-2 UNION structure is accomplished. Based on the results the dimension of 50×50×50 cm and the maximum mass of 50 kg is selected for UPMSat-2 UNION satellite at this stage. For future work FEM analysis and more detailed calculation as well as optimization of the whole structure is foreseen.

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